NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED

May 1942 as
Memorandum Report

GENERALIZED SELECTION CHARTS FOR BOMBERS WITH FOUR 2000-HORSEPOWER ENGINES

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NACA

WASHINGTON

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

GENERALIZED SELECTION CHARTS FOR BOMBERS WITH FOUR 2000-HORSEPOWER ENGINES

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SUMMARY

A study has been made of the characteristics and performanco of current Air Forces airplanes where the material used was that available from flight and wind-tunnel tests and manufacturers' specifications. The information obtained from this study has been used to select families of bombers and to compute their performance.

Performance is represented in a series of charts with coordinates of power loading and wing loading. This type of chart should greatly simplify the selection of the best airplane for a given purpose.

Detailed discussions of the assumptions, of the formulas used in constructing the charts, of the parameters affecting airplane performance, and of the limitations of the charts are given in the appendixes.

INTRODUCTION

The problem of selecting airplane characteristics for particular performance is of great importance from an economic and military point of view. The characteristics which an airplane may have are determined by: (1) the atmosphere in which it must operate, (2) the materials of which it is composed, (3) the characteristics of the fuel and its method of utilization. If there were available materials of greater strength to weight ratio and fuels of more available energy or methods of utilizing the available fuels more efficiently, airplanes would have new characteristics and higher performance.

The airplane designer has the problem of designing the airplane with the proper characteristics or dimensions so that the highest possible performance of a given type is obtained. The problem is to make the optimum choice of (1) power, (2) gross weight, (3) wing area, (4) aspect ratio, (5) load factor, (6) wing thickness, (7) altitude or air density, insofar as a choice may be made.

The proper choice of seven variables to give the highest performance is a tedious job, and a job which due to improvement in material and engines must be done at frequent intervals. Ordinarily a designer relies on experience and availability of certain elements such as engines, propellers, etc., rather than upon a detailed analysis to select each characteristic to the very best advantage. The selection of characteristics in this manner results in the selection of airplanes which are short of the maximum possible performance. Selection of airplanes by experience leads to specifying airplanes without full regard to the physical limitations and possibilities.

Performance charts, such as are presented, give a picture of the relation between the speed, range, climb, and take-off characteristics and relate these characteristics to the principal airplane parameters of wing loading and power loading. These charts enable one to select the parameters which will give a certain type of performance and, even more important, show the impossibility of certain desired performance.

The primary purpose of this report is to show the interrelationships of the performance characteristics. The actual values of the calculated performance are only of secondary importance as long as the trends in performance with the selected parameters are correct. For this reason it is permissible to make simplifications in the methods of analysis that may seem too drastic to the man who has been concerned with accurately determining the performance of each particular existing airplane.

For example, if an airplane is flying to a base 5000 miles out in the ocean and fails to reach the base by 100 miles, the importance of the range shortage means that the airplane and the crew may be lost. However, if the preliminary design of two airplanes shows one to have a range of 5000 miles and the other 4900 miles, the range characteristics are taken to be equal because the broad nature of assumptions in preliminary design does not allow a range estimate to be accurate to 2 percent.

The report should not be interpreted as presenting new methods of performance calculations to supersede accepted methods. In the reading of the report it is hoped that the primary purpose be kept clearly in mind.

In selecting such parameters as wing-thickness ratio, design load factor, and fixed weights, an effort was made to choose values agreeing as closely as possible with modern practice. In the case of drag coefficients, however, two sets of values were chosen, one in agreement with modern airplanes and the other for airplanes having a parasite drag corresponding to simple skin friction.

Suggestions of the Air Materiel Command have been incorporated in the construction of the charts presented. Designers and buyers of airplanes should find charts of this type based on accurate data very useful in the specifying, designing, and operating airplanes.

SYMBOLS

b	wing span
c	minimum specific fuel consumption, pounds per brake horsepower-hour
C_1	coefficient multiplying the distributed load to give the effective distributed load
C_D	drag coefficient
C_{D_o}	parasite-drag coefficient
C_{D_i}	induced drag coefficient
C_L	lift coefficient
D	drag, pounds
e	span factor

F	effective frontal area of the bodies on an airplane, square feet
f	load factor
K	dimensionless wing-weight constant
L	lift, pounds
P	engine brake horsepower
P_c	excess horsepower for climbing
q	dynamic pressure of the air stream $\left(\frac{1}{2}\rho V^2\right)$, pounds per square foot
R	aspect ratio
S	wing area, square feet
s	take-off distance, feet
t	root wing thickness divided by chord
T	net accelerating force (thrust-drag)
V	airplane speed, feet per second
V_c	rate of climb, feet per minute
W	gross weight of the airplane, pounds
W_e	gross weight of airplane less gas and oil and bombs, pounds
W_1	wing weight, pounds
W_2	distributed weight on the wing, pounds
η	propeller efficiency
ρ	air density, slugs per cubic foot
L/D	ratio of lift to drag
W/P	power loading, pounds per horsepower
W/S	wing loading, pounds per square foot

PRESENTATION OF CHARTS

A series of charts is presented (figs. 1 to 8) showing the performance of bombers aerodynamically and structurally about equal to the best produced at the present time and powered by four 2000-horsepower engines supercharged to 25,000 feet. Each chart is made on identical coordinate axes W/P and W/S so that the charts may be superimposed for the easy selection of the most desirable set of performance characteristics. Figure 8 shows a composite of the performance characteristics using only a few of the curves of each characteristic.

Figures 10 to 16 show a similar group of charts for bombers of a greater aerodynamic excellence, their parasite drag being taken as little more than skin-friction drag. Figure 16 shows a composite of the performance characteristics for the low-drag bombers.

Values of maximum L/D for the two groups are given in figures 9 and 17.

The assumptions upon which the charts are built are given in appendix A; the formulas and methods of building the charts are given in appendix B; a discussion of the various parameters affecting airplane performance is given in appendix C; and a discussion of the limitations of the charts is given in appendix D.

Each performance chart represents the performance of a family of airplanes. If several charts representing various types of performance are superimposed to form a composite chart, as in figure 8, then each point on such a chart represents a consistent group of airplane performance characteristics. For example, for a bomber with a wing loading of 37 pounds per square foot and a power loading of 16.2 pounds per horsepower, figure 8 indicates a range of 9000 miles with a 2000-pound bomb load, a take-off distance of 2000 feet, a rate of climb at sea level of 1000 feet per minute, and a high speed of a little over 300 miles per hour at 25,000 feet.

As an illustration of the use of the charts, let it be desired to select a bomber powered by 2000-horsepower engines, with a high speed of 350 miles per hour, a range (with a 2000-pound bomb load) of 9000 miles, a take-off distance not to exceed 2000 feet, and a rate of climb not less than 1000 feet per minute at take-off. On figure 8 the 350-mile-per-hour-speed curve does not intersect the 9000-mile-range curve. Hence, the desired combination is not attainable with this family of airplanes.

However, if an airplane with a range of 7000 miles is acceptable, the specifications become compatible. Any point in the area bounded above by the 350-mile-per-hour curve and below by the 7000-mile curve (fig. 8) represents airplanes which have speeds and ranges greater than the minimum specified. Only area below the 1000-feet-per-minute-climb curve represents airplanes satisfying the climb specification. Only area to the left of the 2000-foot take-off curve represents airplanes satisfying the take-off specification. The area representing bombers fulfilling the specifications is a small triangle bounded by the 350-mile-per-hour curve, the 7000-mile-range curve, and the 2000-foot take-off curve. A power loading of 11.5 pounds per horsepower and a wing loading of 46 pounds per square foot give a slight margin over the minimum specifications. This example is simply illustrative of the use of the charts. Airplanes having other consistent performance characteristics determine other localized areas on the charts.

If the parasite drag can be reduced to skin-friction drag, figure 16 shows the performance trends to be expected. Taking the same values of power loading and wing loading (11.5 and 46) into the chart on figure 16, it will be seen that the take-off distance and rate of climb are the same as in figure 8, or nearly so. The range has been increased to 9000 miles and the speed to over 400 miles per hour.

These examples demonstrate that the specification of airplane performance is intimately bound up with the power loading (gross weight for a given power) and wing loading. If a great range (high power loading) is specified, a high top speed (low power loading) cannot also be specified. Similarly, a high top speed is obtained with a high wing loading and a short take-off distance with a low wing loading. Hence, considerable discretion must be exercised in laying down the specifications for an airplane because of the conflicting demands made by the various performances.

By moving around from point to point of a chart of this type, constructed for a particular engine power and degree of aerodynamic and structural excellence, it becomes possible for the military tactician to select the most suitable combination of performances for any type of mission. This selection automatically determines the proper power loading and wing loading and a considerable portion of the preliminary design.

The optimum wing area for high speed is not nearly so high at high power loadings as at low. Since there is no possible point in selecting a wing loading above the maximum for high speed, it follows that a high wing loading is out of place at a high power loading. This point becomes increasingly evident as the airplanes are supercharged to higher altitudes, for the optimum wing area for speed decreases with increasing altitude. (See equation of optimum wing loading for speed, appendix C.)

In certain cases, as for operation where the operating bases must be set up hastily, the take-off distance may of necessity be small enough to subjugate the other types of performance. There is then considerable logic in moving across the chart along a predetermined line of constant take-off distance in selecting the wing and power loading giving the best compromise of the other performances.

A selection chart constructed for a particular degree of aerodynamic and structural excellence becomes a standard to which the performance of actual airplanes of the same power may be compared. Although it should be realized that different amounts of auxiliary equipment prevent airplanes from being strictly comparable, in general, airplanes with performances widely differing from the performance indicated by a chart are aerodynamically or structurally inferior or superior to the standard.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., May 11, 1942

APPENDIX A

AIRPLANE PARAMETERS SELECTED

Airplane characteristics are subject to evolution. There is a gradual improvement in engines, structures, and aerodynamic design. An effort has been made to base the parameters of the airplanes selected for investigation on the best usage at the time of writing. In estimating weights and drags, liberal use has been made of the information on modern Air Forces airplanes available in the Materiel Command Liaison Office at Langley Field.

The parameters appearing as variables on the selection charts are wing loading and power loading. Other parameters, such as drags and weights, are varied systematically over the charts. Others, such as wing-thickness ratio and aspect ratio, are considered as constants. Appendix A is a discussion of the parameters that are incorporated in the charts but do not appear as chart coordinates.

POWER PLANTS

The bombers are all powered by four 2000-horsepower engines. It is assumed that each requires a nacelle projected frontal area of 25 square feet for adequate housing and the admission of all cooling air. Weight estimates are made to include all auxiliary equipment necessary for full power operation to 25,000 feet. The curves assumed for minimum specific fuel consumption and engine rpm for operation on minimum specific fuel consumption are given in figure 19.

DRAG

Two groups of bombers representing two degrees of aerodynamic refinement have been selected for investigation. Airplanes of one group have a parasite drag equal to that of a modern bomber, one of the best of recent design. This group therefore represents bombers produced at the present state of design progress. The parasite drag of airplanes within this group, based on the total surface area of the airplane, is about 0.0050.

Airplanes of the other group investigated have a parasite-drag coefficient of about 0.0035, based on surface area, or very little more than the turbulent skin-friction drag of aerodynamically smooth surfaces. It is believed that this airplane need not be relegated to the distant future since wind-tunnel tests of a complete model have already demonstrated a design with a parasite drag equal to an equivalent skin-friction drag of 0.0035.

Fuselage and nacelle drag have been based on an "effective" frontal area. This area is constant at 140 square feet for the four nacelles and fuselage. This allows for a fuselage frontal area proportional to the gross weight of the airplane to the two-thirds power. The effective frontal area is taken constant because, as the gross weight increases, the nacelles become effectively submerged in the wing. Figure 18 shows how the nacelle and fuselage areas vary.

The parasite-drag coefficient is made up of the wing, tail, fuselage, and nacelle components. The values chosen to represent the two groups are given in the following table:

Drag Table

Component	Case I		Case II	
	Area based on:	Drag coefficient	Area based on:	Drag coefficient
Wing	Wing area	0.0090	Wing area	0.0070
Tail	Wing area	.0030	Wing area	.0020
Fuselage + nacelles	Effective frontal area	.120	Effective frontal area	.060
Fuselage + nacelles	Wing area	.12F/S	Wing area	.06F/S
Total	$C_{D_o} = 0.0120 + 0.12F/S$		$C_{D_o} = 0.0090 + 0.06F/S$	

SPAN FACTOR

An addition to the parasite and ideal induced drag with increasing lift coefficient is assumed and expressed as an increase in the induced drag. Thus, the induced drag is divided by a "span factor" as in the equation

$$D = C_{D_0}qS + \frac{(W/b)^2}{e\pi q}$$

The value of "e" is taken as 0.8 in this analysis.

PROPELLER EFFICIENCY

It was assumed that a propeller efficiency of 85 percent could be realized. In order to simplify the performance computations, it is assumed that cooling power is proportional to brake power. This assumption makes it possible to take account of the cooling losses by an equivalent reduction of the propeller efficiency. Five percent of the brake power was allowed for cooling, giving an effective propeller efficiency of 80 percent. This value was used in all performance calculations. In order to make a constant value of 80 percent effective propeller efficiency applicable to the range calculations for the condition of maximum L/D and minimum specific fuel consumption, it was necessary to make these computations at sea level. (See the section on propeller selection in appendix C.)

ASPECT RATIO

Figures 20 and 21, computed according to the assumptions used throughout this analysis, show that the effect of aspect ratio on range is not critical over a wide range of aspect ratio. A value of 12 is considered to be reasonable for range and for other types of performance. This value has been used throughout the analysis.

LOAD FACTOR

A design load factor of 4 with the 2000-pound bomb load has been used over the entire chart. This is sufficient to protect against a standard gust of 30 feet per second. Very moderate maneuverability is afforded by this load factor.

WING THICKNESS

A 20-percent wing-thickness ratio at the root chord was used for all the airplanes. This wing is thick enough to keep the wing weight reasonable but not thick enough to cause a high drag or to experience compressibility at maximum speed.

WEIGHT

After a study of Air Forces airplanes, it was assumed that:

1. Fuselage weight is 8 percent of airplane gross weight.

2. Landing-gear weight is 6 percent of airplane gross weight.

3. Tail weight is 10 percent of wing weight.

4. There are certain fixed weights which vary slightly with the gross weight.

Gross weight, lb	60,000	100,000	150,000	200,000
Engines and accessories	18,000	18,200	18,500	18,800
Armor and armament	2,500	3,300	4,100	5,000
Crew and equipment	1,600	2,000	2,000	2,000
Instruments and fixed equipment	700	800	900	1,000
Fixed weights	22,800	24,300	25,500	26,800

5. Weight of fuel system equals 0.55 pound per gallon of gasoline

6. Weight of lubricating system equals 1.25 pounds per gallon of oil.

Sufficient tankage weight is included to obtain maximum range with no bomb load. The tanks are assumed to be carried in the wings.

WING WEIGHT

Wing weight is determined by considerations of strength. An expression equating the internal resisting moment to the external bending moment at the center section gives the following relationship:

$$K = \frac{W - (C_1W_2 + W_1)}{W_1} \times \frac{fR^{3/2}\ S^{1/2}}{t}$$

where K is a dimensionless constant dependent upon:

1. The distribution of lift along the span.

2. The strength weight ratio of the material used in the construction of the wing.

3. The perfection of the design as an efficient weight to strength beam. The higher the K, the more efficient the beam as a weight-carrying structure.

For simple loading conditions, such as those for pursuit airplanes where nearly all of the load is concentrated in the fuselage, it is to be expected that a value of $C_1 = 0$ would approximate the loading conditions. For multiengine bombers, where a large portion of the load is distributed along the wing, a value of C_1 between 0.5 and unity would be expected to approximate the loading condition. The following table shows the values of K computed for $C_1 = 0$ and $C_1 = 1$ for a number of airplanes taken from references 1 and 2 and the files of the Liaison Office of the Materiel Command at Langley Field, Va.

Airplane	Design gross weight, lb	Wing area, sq ft	Root wing thickness ÷ chord	Aspect ratio	Wing weight, lb	Design load factor	Load distribution along wing, lb	K for $C_l = 0$	K for $C_l = 1$
P-36A	5,400	236	0.15	5.9	815	12	------	100,000	------
P-40B	6,700	236	.15	5.9	900	12	------	114,000	------
P-41	6,700	224	.16	5.78	875	12	------	104,000	------
B-15	70,000	2750	.20	8.07	6,600	4.3	45,000	250,000	72,000
B-17B	38,000	1420	.18	7.58	5,554	5.5	18,700	140,000	64,000
B-18A	22,280	965	.15	8.4	3,829	5.5	7,900	134,000	76,500
B-19	140,000	4285	.19	10.68	25,000	4.05	85,000	224,000	58,000
B-24	41,000	1048	.22	11.55	6,774	5.5	18,500	161,000	70,500
B-26	26,500	602	.17	7.03	2,900	5.5	------	114,000	------
B-32	95,500	1422	.23	12.81	12,500	5.5	60,000	275,000	76,500

For the purpose of this analysis, a value of $K = 100,000$ and a value of $C_1 = 0.85$ were used on the basis of the study of existing airplanes. To solve this equation for wing weight if the value of the load to be carried in the wings is as yet unknown, W_2 may be conveniently expressed as the gross weight less the weight of the fuselage and the weight carried by the fuselage (including the tail surfaces) less the wing weight.

Figure 20 shows the way structural weight and weight of gas, oil, and bombs vary with wing loading and power loading for the assumptions outlined in this appendix.

APPENDIX B

METHODS OF COMPUTATION

There are several types of performance for which an airplane may be designed, such as range, speed, take-off, and climb. Each of these will be considered and the formula presented from which the computations have been made.

It would be almost impossible to construct general charts if each possible airplane described by the chart was computed with the detail which an airplane designer uses for one airplane. It is thus necessary to make estimates of drag, weight, propeller efficiency, cooling power, etc., which are either constant or vary in a systematic way over the possible range of parameters covered in the chart.

This section will be devoted to presenting the formula and introducing the necessary simplifying estimates for the construction of the charts.

RANGE

The range of an airplane may be computed from the Breguet formula by a step-by-step method as suggested by Diehl.

$$\text{range} = 375 \frac{\eta}{c} \frac{L}{D} \log_e \frac{W_1}{W_2}$$

where

η propeller efficiency

c specific fuel consumption

L/D lift to drag ratio

W_1 airplane weight at the beginning of an increment of range

W_2 airplane weight at the end of an increment of range

The application of this formula to a particular airplane is simply a matter of selecting the proper values of the variables for each increment of range considered. For a particular airplane the value of L/D depends on the flying attitude, the value of c depends on the power output and rpm of the engines, and the propeller efficiency depends on the adaptability of the propeller.

A computation of the maximum possible range for a given airplane requires a rigorous analysis of the variation in the expression $\frac{\eta \, L/D}{c}$. However, when it is desired to only give a picture of how the range varies with large changes in the parameters of the airplane, such as wing loading or power loading, then certain simplifications to the calculations are permissible.

For the purposes of this report it was assumed that η remains constant at 80 percent throughout the flight (see section on propeller efficiency), the airplane is always flown at maximum L/D (see section on maximum L/D); and that the value of c only varies with the engine power (see section on specific fuel consumption). Oil consumption has been accounted for by assuming the oil consumption is equal to 5 percent of the fuel consumption. This assumption is the equivalent of introducing a multiplier of 1.05 in the denominator of the range equation.

The brake horsepower required to fly the airplane is

$$P = \frac{DV}{550}$$

$$= \frac{1}{550}\sqrt{\frac{2}{C_L}}\,\frac{S}{L/D}\left(\frac{W}{S}\right)^{3/2}$$

Introducing the conditions for max L/D, that induced drag is equal to the profile drag of the airplane,

$$C_D = C_{D_o} + \frac{C_L^{\,2}}{e\pi R} = 2C_{D_o}$$

$$C_L = \sqrt{e\pi R C_{D_o}}$$

Dividing C_L by C_D we obtain

$$\max L/D = \frac{1}{2}\sqrt{\frac{e\pi R}{C_{D_o}}}$$

At max L/D the power equation becomes

$$P = \frac{2^{3/2} C_{D_o}{}^{1/4} S}{550\eta\rho^{1/2} (e\pi R)^{3/4}} \left(\frac{W}{S}\right)^{3/2}$$

Knowing the power which must be developed by the engines for level flight at max L/D, a curve of specific fuel consumption is consulted to obtain the value of c in the range equation.

RANGE REDUCTION

The range reduction is taken from the curves of flying weight versus range obtained in the process of range computation. The range reduction is obtained on the assumption that the bombs are dropped at a distance equal to one-half the range.

MAXIMUM SPEED

The maximum speed was computed from the basic relations:

$$P = DV/\eta$$

$$D = \left(C_{D_o} + \frac{C_L{}^2}{e\pi R}\right) \frac{\rho}{2} SV^2$$

$$C_L = \frac{W}{\frac{\rho}{2} SV^2}$$

These formulas combine to give

$$W = \sqrt{\frac{e\pi R}{2} \rho SV \left(550\eta\ P - \frac{C_{D_o}}{2} \rho SV^3\right)}$$

By substituting values of V and S in the above equation, the value of W is computed and curves of constant speed are obtained as in figure 1.

RATE OF CLIMB

The rate of climb is determined at max L/D by the excess power available for climbing over that required for level flight. The general expression for rate of climb is

$$V_c = \frac{\eta P_c \; 33000}{W}$$

where

$$P_c = P - \frac{1}{550\eta}\sqrt{\frac{2}{\rho C_L}}\,\frac{S}{L/D}\left(\frac{W}{S}\right)^{3/2}$$

Substituting the expressions for max L/D

$$\max\frac{L}{D} = \frac{1}{2}\sqrt{\frac{e\pi R}{C_{D_o}}}$$

and

$$C_L = \sqrt{e\pi R C_{D_o}}$$

it follows that

$$V_c = 33000\eta\left[\frac{P}{W} - \frac{2\sqrt{2}\,C_{D_o}{}^{1/4}}{550\eta\rho^{1/2}\,(e\pi R)^{3/4}}\left(\frac{W}{S}\right)^{1/2}\right]$$

TAKE-OFF RUN

The take-off run is calculated assuming a level field and no wind. Propeller efficiency is assumed to vary linearly from zero at the beginning of the run to 80 percent at 90 miles per hour and to remain constant at 80 percent above 90 miles per hour. In order to simplify the calculations, rolling friction and air resistance during take-off are accounted for by assuming this resistance is equal to

10 percent of the propeller thrust. The lift coefficient at the instant of take-off is taken as $C_L = 1.3$. The distance to clear an obstacle is not included in the distance given.

The basic equation for computing the take-off distance is

$$s = \int_0^{V_{to}} \frac{WVdV}{gT}$$

where V_{to} is the take-off speed, feet per second. For the assumptions just stated and if the take-off speed is less than 90 miles per hour, this equation integrates to

$$s = 3.35 \frac{W}{P} \frac{W}{S}$$

If the take-off speed is above 90 miles per hour, the equation becomes

$$s = 30.1 \frac{W}{P} + 0.43(W/S)^{3/2} \frac{W}{P}$$

A comparison of the above method with the more exact method used by the Materiel Command, taking into account ground friction and aerodynamic drag, shows that the take-off distance as computed in this report is slightly too long for the light wing loadings and is slightly too short for the very high wing loadings. For the comparison made, the two curves cross in the neighborhood of 70 pounds per square foot wing loading. Because the more exact method required a graphical integration of each point on the chart and the method used in this report requires only the solution of an equation, there is a vast difference in the labor required by the two methods. The method used seemed justified for use as an indication of the variation of take-off distance with the other airplane parameters.

APPENDIX C

DISCUSSION OF PARAMETERS AFFECTING PERFORMANCE

Aspect Ratio

The aspect ratio used in the design of an airplane is determined by a compromise between structural weight and induced drag. High aspect ratio gives high structural wing weight and low induced drag. For a given gross weight, the increase in structural wing weight decreases the fuel load and thus the range. The decrease in induced drag resulting from an increase in aspect ratio increases the distance traveled on a given fuel load. A balance between these two factors determines the best aspect ratio for maximum range.

A comparison of the aspect ratio selected for pursuit airplanes and four-engine bombers immediately reveals that the pursuits have a lower aspect ratio. This has come about because the pursuit airplanes are designed with high load factors, concentrated loads in the fuselage, and thin wings for compressibility requirements. All of these factors increase the relative importance of wing weight. The bombers are designed with low load factors; a large part of the load is distributed along the wing, and thicker wings are used than on pursuit airplanes. These factors tend to minimize wing weight. In this case maximum range is obtained with a relatively high aspect ratio.

The preceding illustration serves to show the extent to which the optimum aspect ratio depends on the parameters of load factor, the load distribution, and the wing-thickness ratio. Figures 21 and 22, computed according to the assumptions of appendix A, show that the optimum aspect ratio increases with an increase in wing loading. As the wing loading is increased, the induced drag becomes of increased importance and the optimum aspect ratio is increased. It will be noted, however, that the curves of range versus aspect ratio are very flat, and the effect of aspect ratio on bomber range is not critical over a wide range of aspect ratio.

The analysis of this paper assumes that the wing weight is a function of the bending moments in the wing.

This assumption may not be true for a high-speed multi-engine bomber with a large part of the load distributed throughout the wing because the torsional rigidity necessary to keep the wing free from flutter troubles may give the most serious design condition. Such a design condition may force the selection of a lower aspect ratio for the airplane.

Load Factor

Performance is vitally affected by design load factor. If a bomber were designed with a load factor similar to that of a pursuit airplane, its range and load-carrying capacity would be seriously reduced. The low load factors used for heavy bombers require that maneuvers be restricted but give a low structural weight that permits a large useful load of bombs and fuel. The extent to which the load factor may be reduced is limited by the gust loads encountered in flight.

The effect of design load factor on performance accounts for the variety of alternate loading conditions and corresponding load factors which are considered in airplane specifications. For a given airplane, the disposition of the load about the airplane determines the maximum operating or "limit" load factor. For example, the design load factor for an airplane may be 4 for a loading condition of one 2000-pound bomb and the remainder of the load as gasoline distributed along the wing span. However, if 15,000 pounds of bombs are carried in the fuselage and the gasoline load is decreased to give the same take-off weight, the load factor may be reduced to 3 by this loading condition.

In the latter case the bomber has a short-range mission. In reality there is nothing in such a mission which should permit a lower load factor than a long-range scouting operation. Load factors used in practice are not entirely logical, but rather are a result of using a given type of airplane for different types of duty.

A point worthy of consideration is the torsional rigidity of the wing. The flutter tendency of the wing depends on the relation between its bending and torsional rigidity. High aspect ratio and increasing speeds place increasing importance on the flutter problem. The structural weight of high-speed-bomber wings may eventually depend more on flutter than on bending and the load factor.

Propeller Selection

If the high speed and maximum range of an airplane are both to be obtained at the same altitude, it is necessary to select a propeller that is a compromise between these conditions. For the maximum-range condition, a large propeller diameter is required to absorb the engine power at the rpm required for minimum specific fuel consumption. This large diameter increases the propeller weight, increases the weight of the landing gear, and reduces the L/D of the propeller section for the high-speed operating condition. If the optimum-range propeller is selected, it may penalize the effective high-speed efficiency as much as 5 percent. However, if the optimum high-speed propeller is selected and the maximum-range condition of flight is neglected, the propeller will stall at maximum L/D and minimum specific fuel consumption, giving a serious reduction of range.

If the high-speed design is for high altitude and the maximum-range condition is desired for low altitude, then a given propeller may be optimum for both conditions of flight and no compromise is necessary. The high-speed condition at 25,000 feet, as used in this report, gives propeller operating conditions that are nearly identical with the maximum L/D condition for minimum specific fuel consumption at sea level. For this reason the range has been computed for sea level throughout the report. The assumption of 85 percent propeller efficiency for the conditions of this report closely approximates the true efficiency. If the range had been computed for 25,000 feet altitude, it would have been necessary to make an analysis of the expression $\frac{\eta\ L/D}{c}$ for each airplane and each loading condition in order to get the maximum range, because operation at maximum L/D, at minimum specific fuel consumption, and at maximum propeller efficiency would have been impossible. The range at sea level is the maximum range obtainable with no wind. The range remains constant as the altitude increases up to the altitude at which the increased speed requires too much power for operation at minimum specific fuel consumption (see section on specific fuel consumption) or the increased altitude loads the propeller up until some of the propeller sections stall.

The magnitude of the change in propeller efficiency due to compressibility effects for flight conditions is not well defined at the present time. Some preliminary data indicate that the conditions of flight differ considerably from those in a wind tunnel. These data indicate that the loss in efficiency due to compressibility for the conditions of the test was much less than would be expected from tunnel tests. These results might be interpreted as an extension of the subsonic range of flight or might be interpreted as an indication of the possibility of supersonic flow without compressibility shock. The explanation of flight test results on propellers operating in the range where compressibility losses would be expected from wind-tunnel tests is one of the most important problems for present-day research since, for high-speed airplanes operating at high altitude, the entire airplane design is critically dependent upon compressibility considerations.

Granting the incompleteness of the knowledge of compressibility effects, certain things may be said regarding the change in compressibility conditions with operating condition. For the high-speed condition of flight, the adverse effects of compressibility are always less as the altitude is decreased. The increase in air density, as the altitude is decreased, lowers the propeller section lift coefficient and thus the local velocity over the propeller sections. The higher air temperature at low altitude increases the speed of sound. These two considerations are sufficient to change the operating conditions of a propeller so that it may be in serious trouble over the entire radius at 25,000 feet and be completely free of trouble at sea level.

Specific Fuel Consumption

Figure 19 shows how the minimum specific fuel consumption and the rpm for minimum specific fuel consumption vary with horsepower for an existing 2000-horsepower engine. The curve of minimum specific fuel consumption is taken from a family of test curves for this engine giving the variation of specific fuel consumption with rpm for various constant horsepowers. An envelope of the minimum points of this family of test curves yields the two curves of figure 19. The significance of the rpm curve is discussed in the section on propeller selection.

To obtain maximum range for flight at maximum L/D and constant propeller efficiency, operation on minimum specific fuel consumption is necessary. If other condiditions permit, it is desirable to operate a particular airplane on powers corresponding to flat portion of the minimum specific fuel-consumption curve (below 800 horsepower, fig. 19) where the values are lowest.

For operation at maximum L/D the speed increases as the altitude is increased and the power required to fly increases in direct proportion to the speed. It follows that, for a given airplane and the engine used in this analysis, an altitude will eventually be reached where the specific fuel consumption will begin to rise because the engine power exceeds 800 horsepower. For the case of the heavily loaded bomber that required 800 horsepower or more at sea level to fly at maximum L/D, the range will decrease with altitude as the power increases and the minimum specific fuel consumption increases. Thus, insofar as the limits of engine economy are concerned, the same range as obtained at sea level may be obtained up to the altitude requiring 800 horsepower per engine.

Maximum Lift to Drag Ratio

For a constant fuselage and nacelle frontal area the maximum L/D is, in general, increased by increasing the wing area. Then, in order to balance induced and parasite drag, the speed at maximum L/D is reduced. The top speed is also reduced because of the increased skin-frictional area.

Increasing maximum L/D is one method of increasing the range. This may be accomplished by increasing the wing area to the point where the increase in structural weight for a fixed gross weight cuts into the fuel capacity to offset the increase in L/D. These points are the minimum points on the constant range curves on any of the range charts.

If, on the other hand, maximum L/D is increased by improving the aerodynamic cleanliness of the airplane, not only is the range increased but the speed at maximum L/D, the top speed, and the speed for any given engine power are also increased. An idea of the increase of range and top speed obtainable by this method may be had

by comparing the charts for the bombers with
$C_{D_o} = 0.0120 + \frac{16.8}{S}$ with those for bombers with
$C_{D_o} = 0.0090 + \frac{8.4}{S}$.

The climbing speed for a given power loading always increases as the L/D ratio of an airplane is improved.

If the L/D is increased by an increase in aspect ratio at constant power loading, the rate of climb and high speed will be improved.

Power Loading

Obviously, the top speed and rate of climb decrease with increasing power loading, wing loading remaining constant. An inspection of the range charts shows, on the contrary, that range increases markedly with increase in power loading. This is because the proportionate decrease of weight of engines and accessories and the resulting increase in fuel capacity is the predominating factor. The increase in range with power loading is rapid until the power loading reaches the point where the specific fuel consumption of the engines begins to rise. From this point on the range increases less and less rapidly up to the limiting condition of full power required to fly at maximum L/D.

A cruising speed defined by a given percentage of rated power will, of course, decrease with increased power loading either with constant wing area or wing loading. However, the speed at maximum L/D will not be inherently changed unless something is done at the same time to change the parasite-drag coefficient of the airplane.

It can be argued that as the power loading is increased the wing loading must be decreased sufficiently to maintain a reasonable take-off run, and a lower speed at maximum L/D is the result. This effect is more properly charged to the effect of wing loading.

Wing Loading

An inspection of the performance charts shows that for a given power loading (or gross weight) there is an optimum wing loading for high speed. The optimum wing loading is seen to become larger with decreasing power loading and increasing speeds. The optimum occurs for $\left(C_{L_{H.S}}\right)^2 / e\pi R$ is equal to the profile-drag coefficient of the wing and tail. In this analysis tail-surface areas have been taken as proportional to the wing area and, consequently, tail drag acts as an increase in wing profile drag. The following equation is a solution for the optimum wing loading for high speed:

$$\frac{W}{S} = \frac{\rho}{2} V_{H.S.}{}^2 \sqrt{e\pi R C_{D_o}}$$

In this equation C_{D_o} is the profile-drag coefficient of the wing plus any other drag effect varying directly with wing area (as tail drag in this report).

The charts also show that for a given power loading there is an optimum wing loading for range. The optimum wing loading increases with power loading and increases slightly with bomb load. The value of the optimum wing loading is rather moderate, ranging roughly from 20 to 60 pounds per square foot.

The rate of climb decreases slowly with increasing wing loading while take-off distance increases very rapidly with increasing wing loading.

Power Per Engine

The optimum amount of power per engine from an aerodynamic point of view has recently become a debatable question because of the high power that is now available per engine and the high altitude at which this power is maintained. The combination of high power and high altitude demands a large propeller to absorb the power efficiently and, consequently, the weight and complication of the propeller are strong factors tending to limit the unit engine power. An adequate treatment of this problem would require a separate paper in order to survey the field, but an idea of some of the factors involved may be obtained from the following table:

		600 HP		1050 HP		2000 HP		4000 HP	
		SEA LEVEL	25,000 FT ALT	SEA LEVEL	25,000 FT ALT	SEA LEVEL	25000 FT ALT	SEA LEVEL	25000 FT ALT
ENGINE WEIGHT, LBS.		920	920	1440	1440	2240	2510	4000	4600
WEIGHT, LBS/HP		1.53	1.53	1.37	1.37	1.12	1.25	1.00	1.15
FRONTAL AREA, SQ. FT.		14.4	14.4	12.5	12.5	15.0	15.0	25.0	25.0
HP PER SQ.FT. FRONTAL AREA		41.7	41.7	84.0	84.0	133.2	133.2	160.0	160.0
TURBO SUPERCHARGER INSTALLATION WT., LBS					550		900		1500
WEIGHT, LBS/HP					.52		.45		.37
ENGINE PLUS SUPERCHARGER WT., LBS/HP					1.89		1.70		1.52
200 MPH	PROPELLER DIAM., FT	9.35	15.10	10.14	16.62	12.30	20.40	13.72	23.10
	NUMBER OF BLADES	2	2	3	3	4	4	6	6
	PROPELLER EFFICIENCY	.882	.888	.869	.876	.860	.868	.824	.839
	PROPELLER WEIGHT, LBS	84	353	161	710	383	1750	797	3800
	WEIGHT, LBS/HP	.14	.59	.15	.68	.19	.88	.20	.950
	TOTAL WEIGHT, LBS/HP	1.67		1.52	2.57	1.31	2.58	1.20	2.47
300 MPH.	PROPELLER DIAM. FT	7.90	13.23	8.40	14.28	10.22	17.22	11.56	19.48
	NUMBER OF BLADES	2	2	3	3	4	4	6	6
	PROPELLER EFFICIENCY	.907	.910	.898	.902	.890	.894	.868	.874
	PROPELLER WEIGHT, LBS	51	237	92	450	219	1040	477	2280
	WEIGHT, LBS/HP	.09	.40	.09	.43	.11	.52	.12	.57
	TOTAL WEIGHT, LBS/HP	1.62		1.46	2.32	1.23	2.22	1.12	2.09
400 MPH.	PROPELLER DIAM., FT	7.14	12.05	7.70	13.06	9.29	15.87	10.75	18.10
	NUMBER OF BLADES	2	2	3	3	4	4	6	6
	PROPELLER EFFICIENCY	.910	.908	.904	.900	.896	.894	.879	.886
	PROPELLER WEIGHT, LBS.	38	179	71	344	165	827	383	1830
	WEIGHT, LBS/HP	.06	.30	.07	.33	.08	.41	.10	.46
	TOTAL WEIGHT, LBS/HP	1.59		1.44	2.22	1.20	2.11	1.10	1.98

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All of the assumptions for the 4000-horsepower engines are extrapolated values and consequently are subject to large inaccuracies, but the propeller calculations are representative of current practice.

It was assumed that the number of blades increased with the power in order to keep the propeller diameter and weight as low as possible for the high-power engines. The weight of the propeller that would absorb 4000 horsepower at 400 miles per hour and 25,000 feet altitude is seen to be 1830 pounds, or a weight per horsepower of 0.46 pound per horsepower.

APPENDIX D

DISCUSSION OF LIMITATIONS OF THE CHARTS

Maximum Speed at 25,000 Feet Altitude

The actual values of speed are very dependent upon the assumptions of drag, aspect ratio, propeller efficiency, and altitude, but are independent of the assumptions on weights or load factors. The trends of speed versus W/P and W/S are correct providing that the same aerodynamic cleanness is obtained on all bombers represented on the chart. The primary use of the speed chart by itself is to afford a means of estimating the effect of varying the gross weight or the wing area, or both, on the speed of a proposed airplane.

Range at Sea Level

The range as calculated is the range which can be obtained carrying the bomb load half-way. The calculation was made for sea level in order to avoid trouble with overloading the propeller due to the low rpm required for minimum specific fuel consumption at small powers. The range is applicable to any higher altitude that does not decrease the ratio of η/c. The variation of this ratio with altitude is dependent upon the power required and the propeller design. The larger the propeller, the higher altitude at which the maximum ratio can be obtained.

The possible range of these airplanes at the design altitude under service conditions of operation is of the order of two-thirds to three-fourths of the values shown on the charts.

Rate of Climb at Sea Level

The rate of climb of an airplane is primarily dependent upon the coordinates of W/P and W/S. The rate-of-climb formula shows that W/P and W/S are the primary variables in the formula and that the C_{D_o} comes in only as the fourth root. This means that an estimation of the rate of climb of any modern airplane may be obtained by the use of the chart with the coordinates of W/P and W/S.

The use of 80 percent propeller efficiency for the rate-of-climb condition at sea level is justified providing that the propeller is correctly designed for the high-speed condition at 25,000 feet. The high altitude with high speed, low density, and low speed of sound imposes a more severe propeller condition than the low altitude with low speed, high density, and high speed of sound.

Take-Off Run

The take-off chart (fig. 7) is also drawn using the coordinates W/P and W/S of the chart and consequently may be applied directly to all airplanes. The take-off distance of an airplane depends on many things, such as the type and condition of the runway, the lift coefficient maintained by the pilot during the run and at the instant of take-off, and the average propeller efficiency during the run. The assumptions made for propeller efficiency during the take-off run are for the recommended propeller. The manner of accounting for friction is very approximate and tends to favor the heavy planes relative to the lighter ones.

CONCLUDING REMARKS

The foregoing discussion of the charts shows their use and limitations. These charts are simply illustrative of a systematic method of presentation which allows the selection of an airplane in a manner so that one may see the complete compromise which is being made. Each airplane designer probably will have other assumptions which he will wish to use in building charts of his own.

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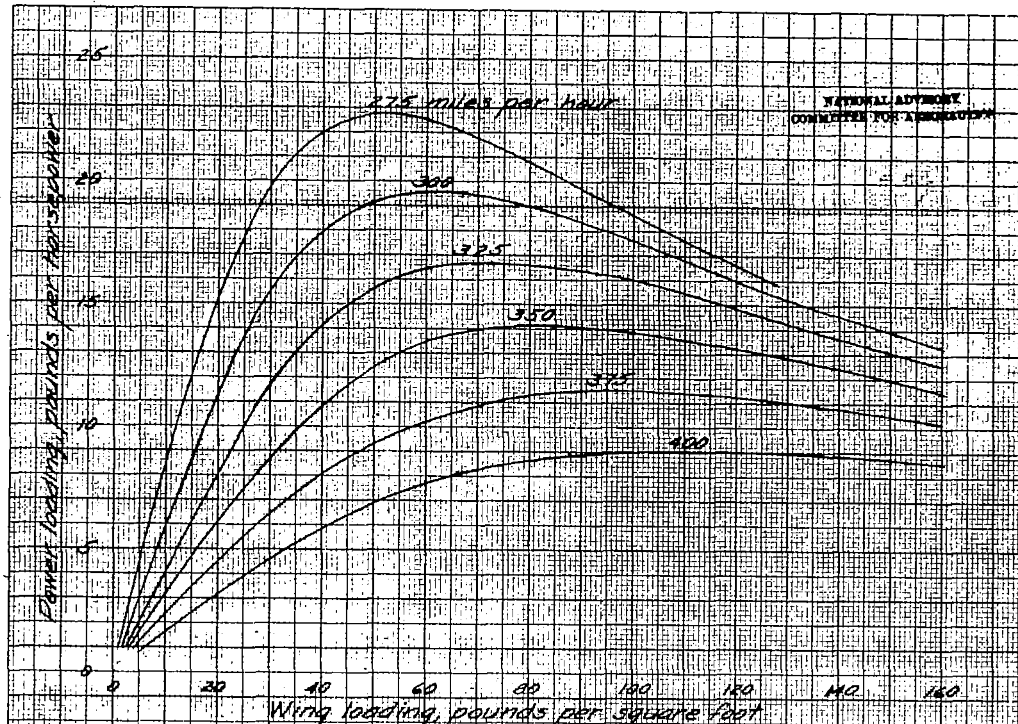

Figure 1- High speed at 25000 feet $C_{D0} = .0120 + .12\,F/S$

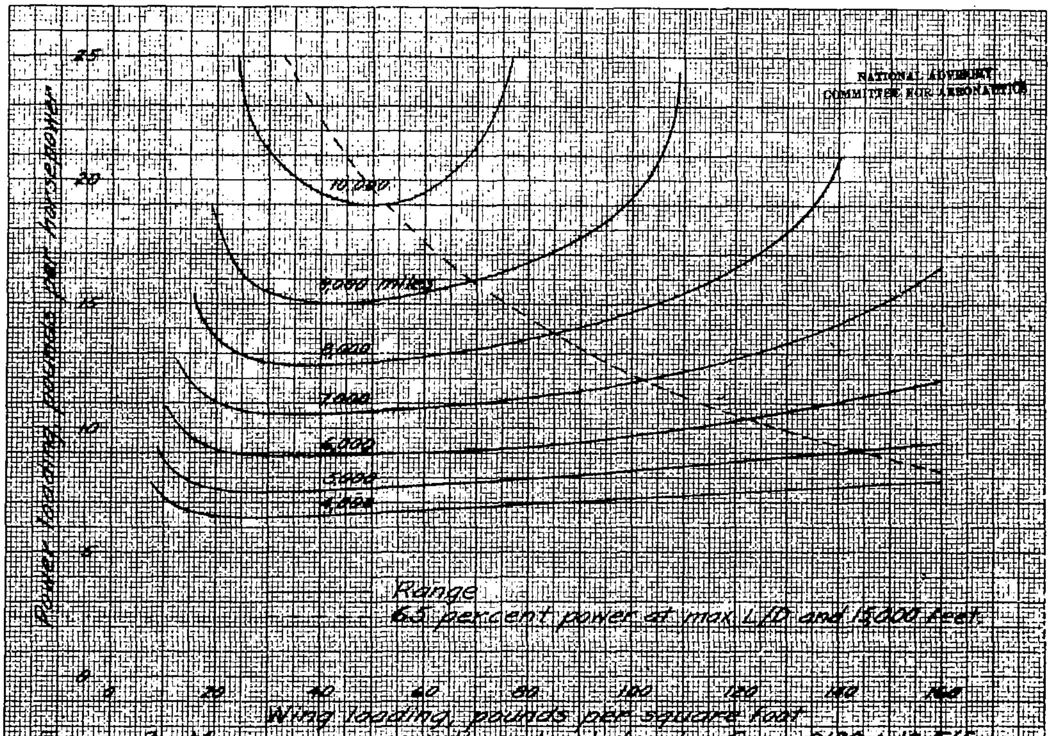

Figure 2- Maximum range with no bomb load $C_{D0} = .0120 + .12\,F/S$

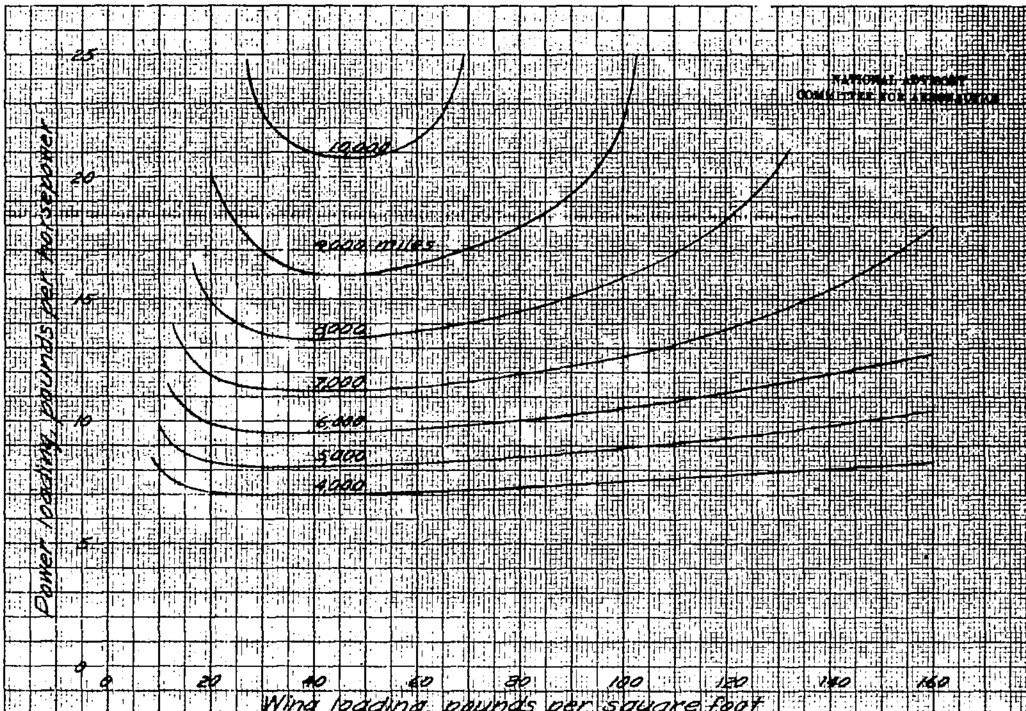

Figure 3 - Maximum range with a 2,000 pound bomb load. C_{D0} = .0120 + .12 F/S

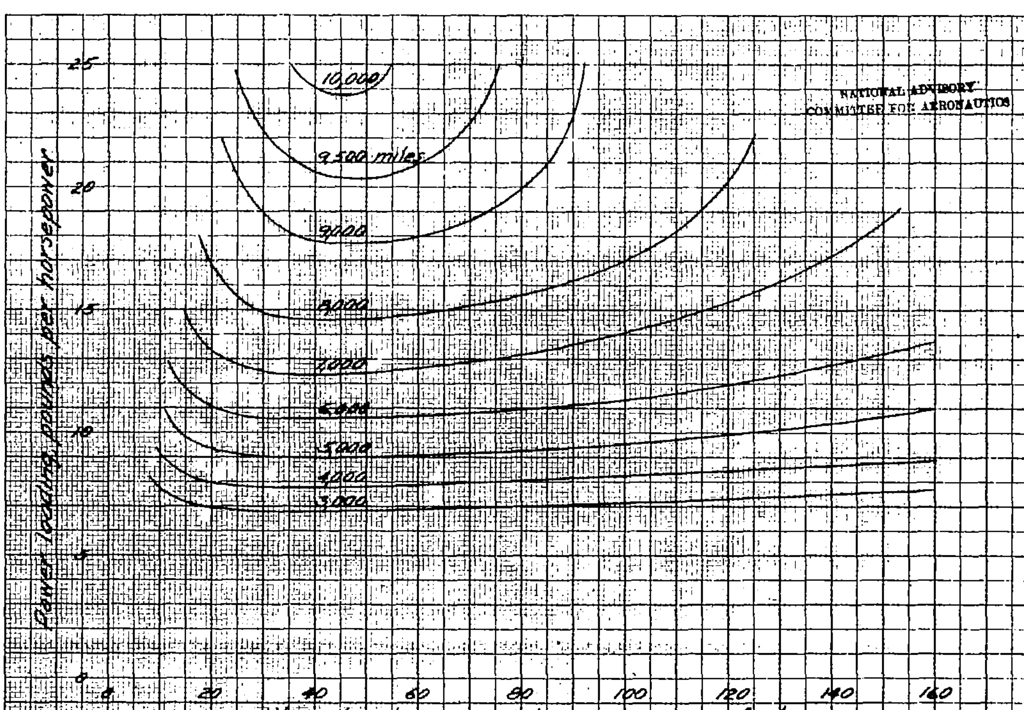

Figure 4 - Maximum range with a 5,000 pound bomb load. C_{D0} = .0120 + .12 F/S

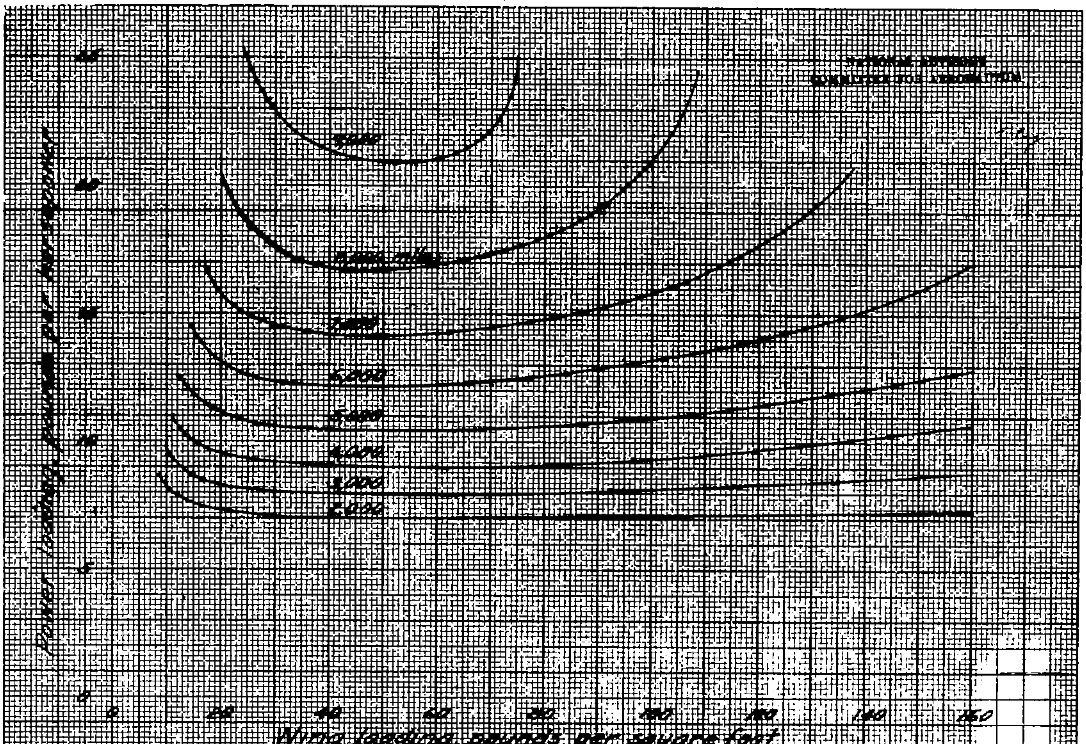

Figure 5 - Maximum range with a 10,000 pound bomb load. C_{D0} = .0210 + .12 F/S

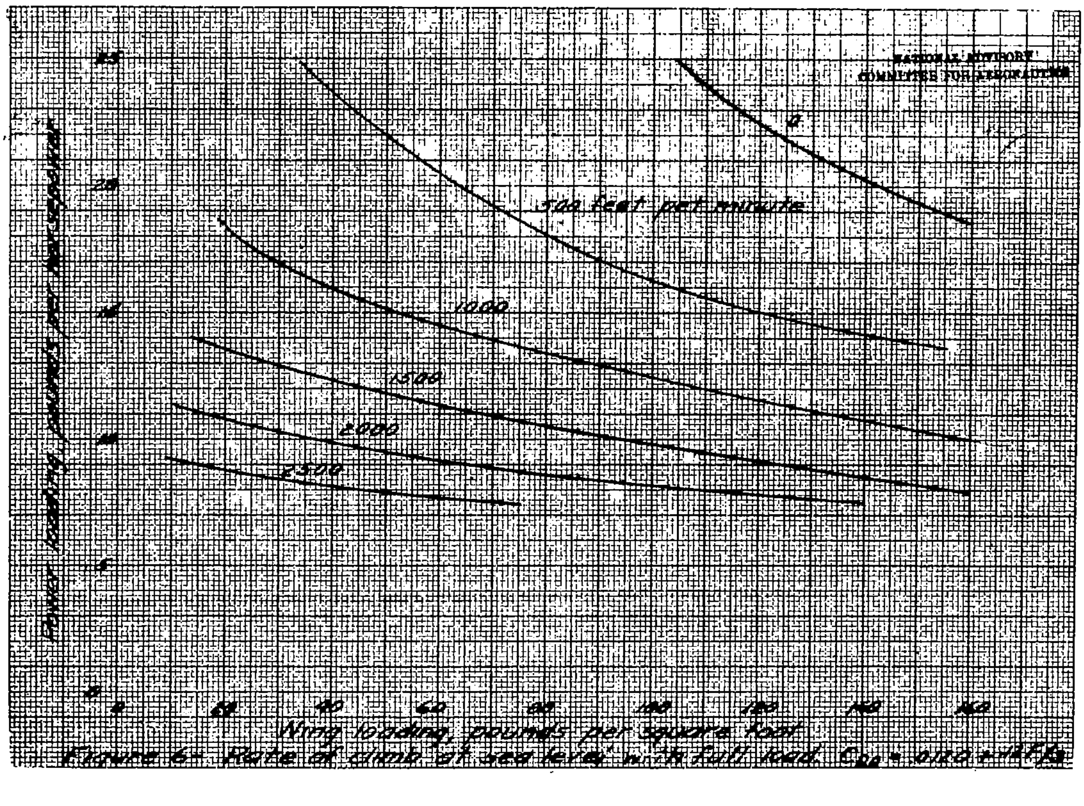

Figure 6 - Rate of climb at sea level with full load. C_{D0} = .0120 + .12 F/S

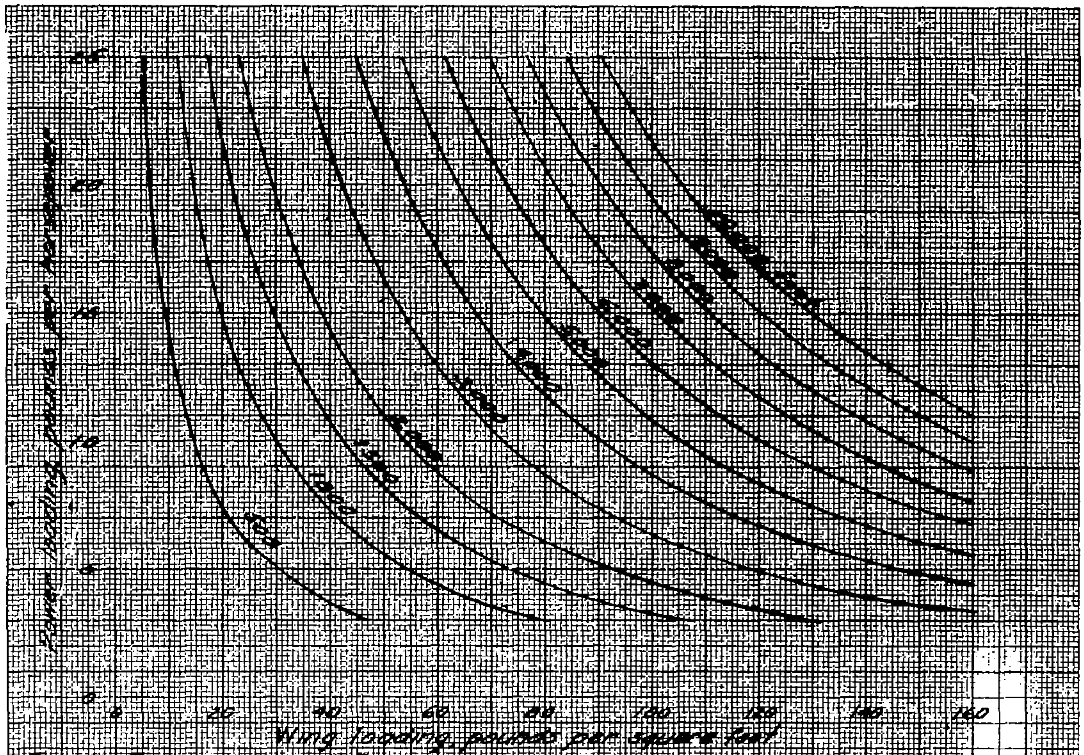

Figure 7.- Curves of constant take-off run.

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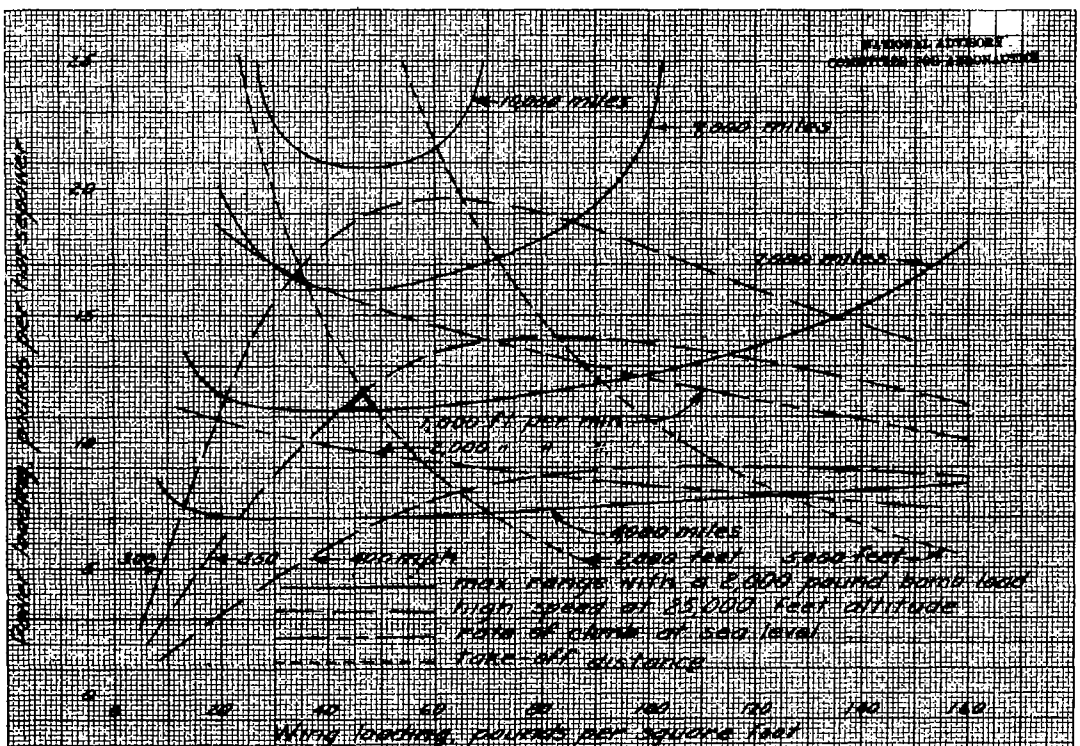

Figure 8.- Performance selection chart. C_{D_0} = .0120 + .12 F/S

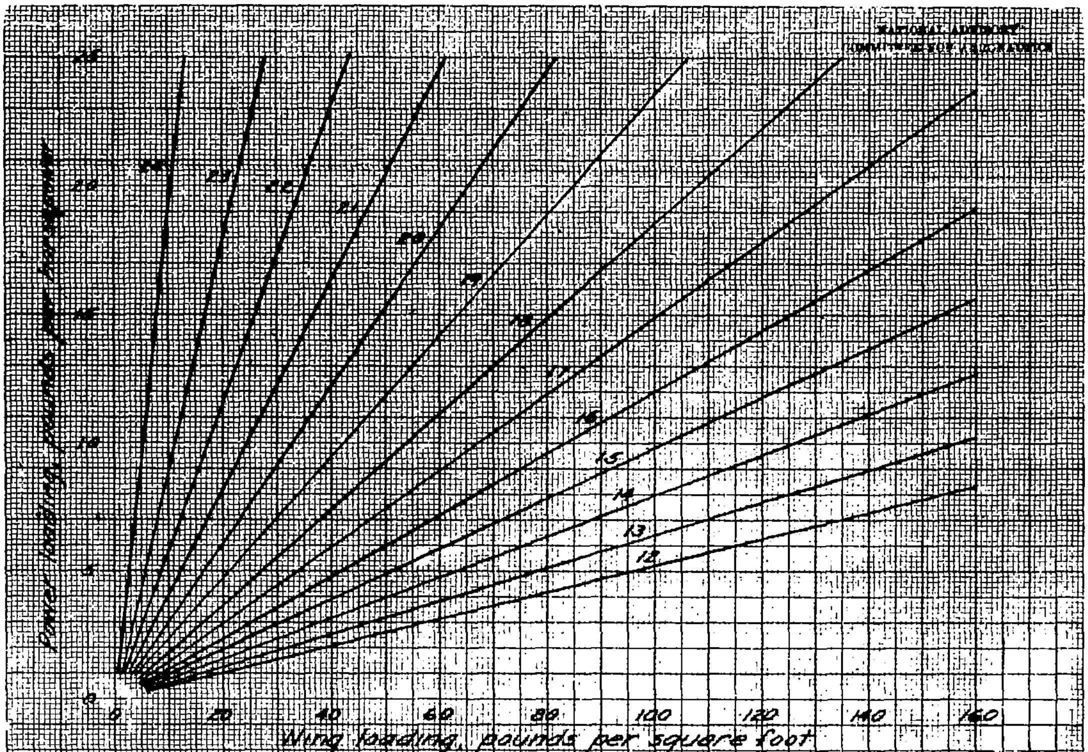

Figure 9- Maximum lift-to-drag ratio. C_{D0} = .0120 + 0.12 F/S

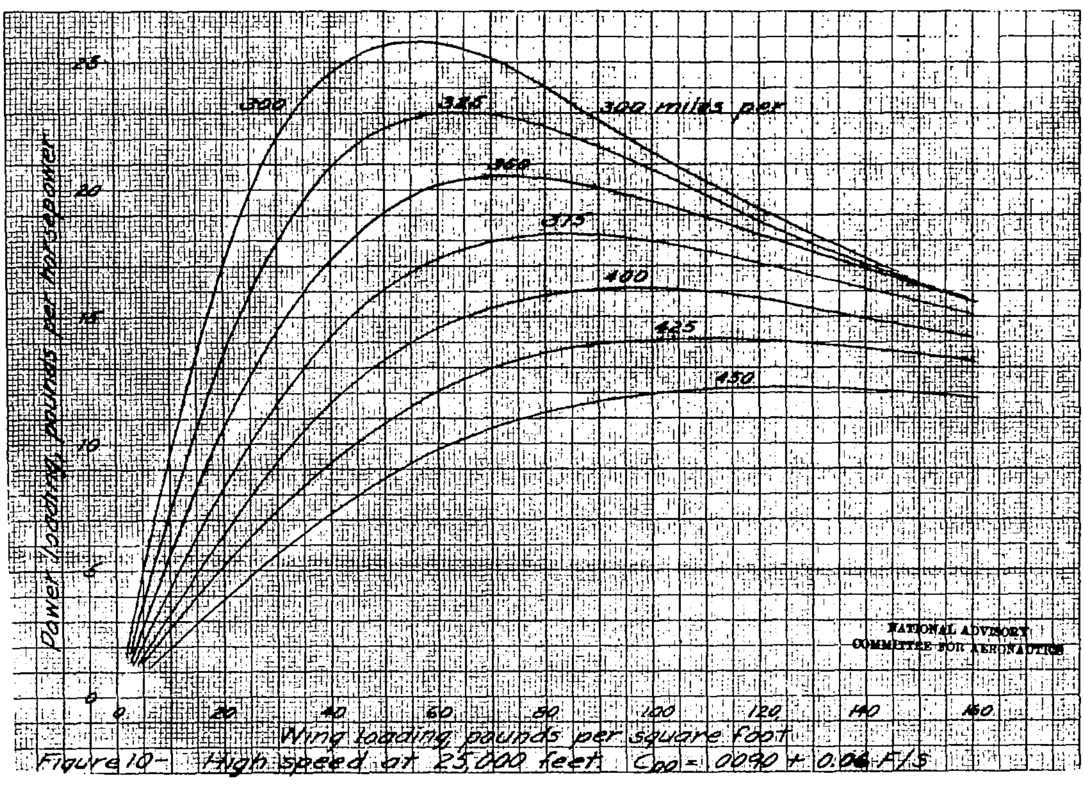

Figure 10- High speed at 25,000 feet. C_{D0} = .0090 + 0.06 F/S

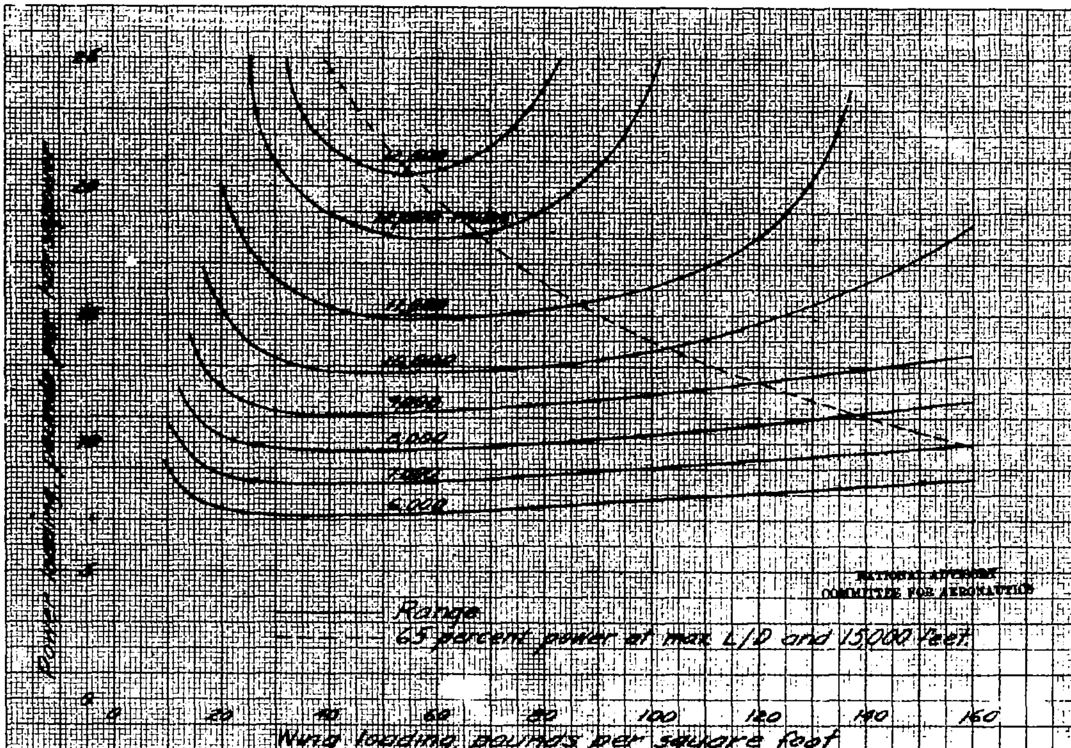

Figure 11 - Maximum range with no bomb load. C_{D_0} = .0090 + .06 F/S

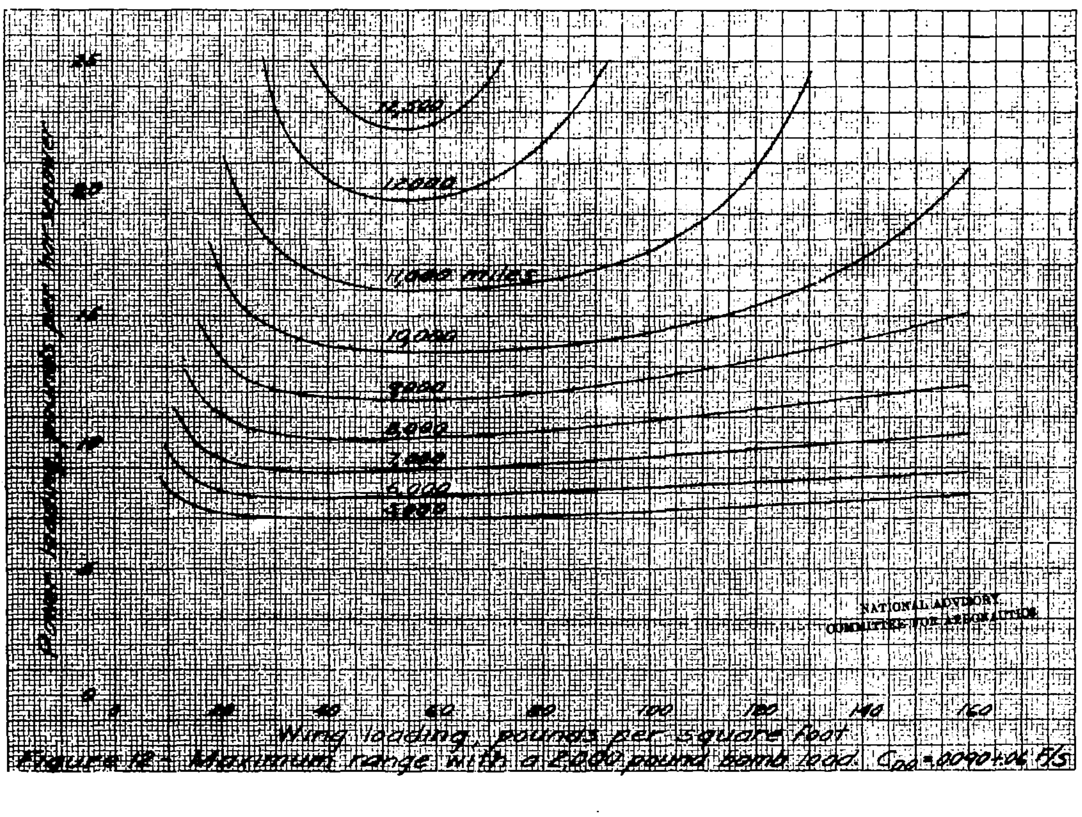

Figure 12 - Maximum range with a 2,000 pound bomb load. C_{D_0} = .0090 + .06 F/S

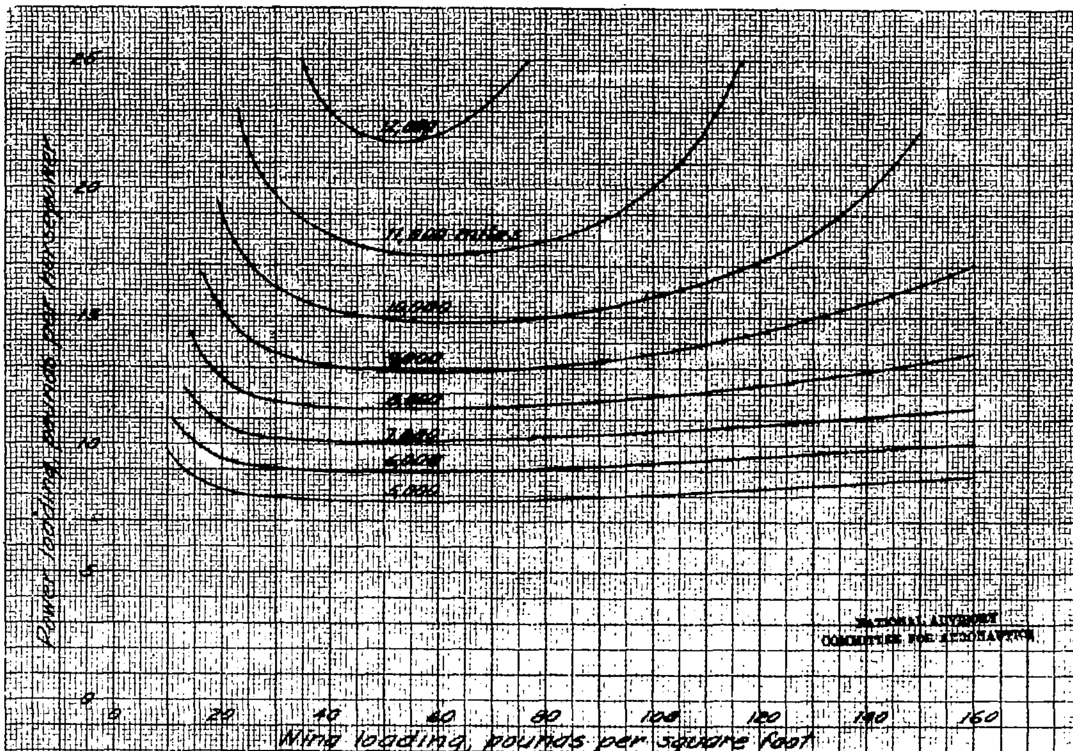

Figure 13 - Maximum range with a 5,000 pound bomb load. C_{D_0} = .0090 + .06 F/S

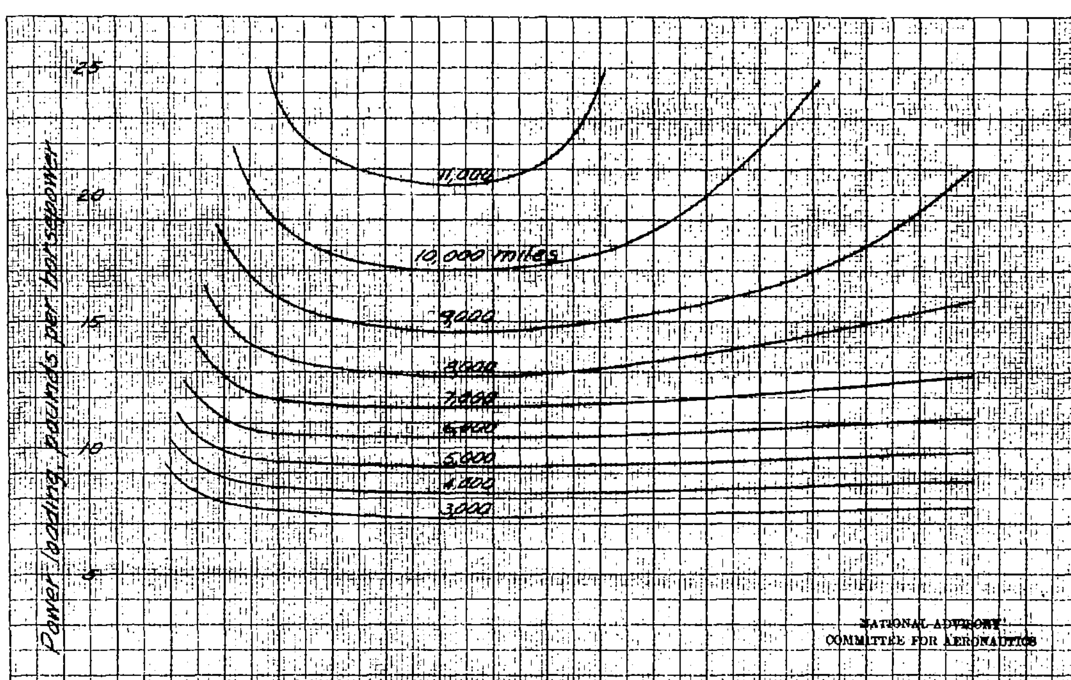

Figure 14 - Maximum range with a 10,000 pound bomb load. C_{D_0} = .0090 + .06 F/S

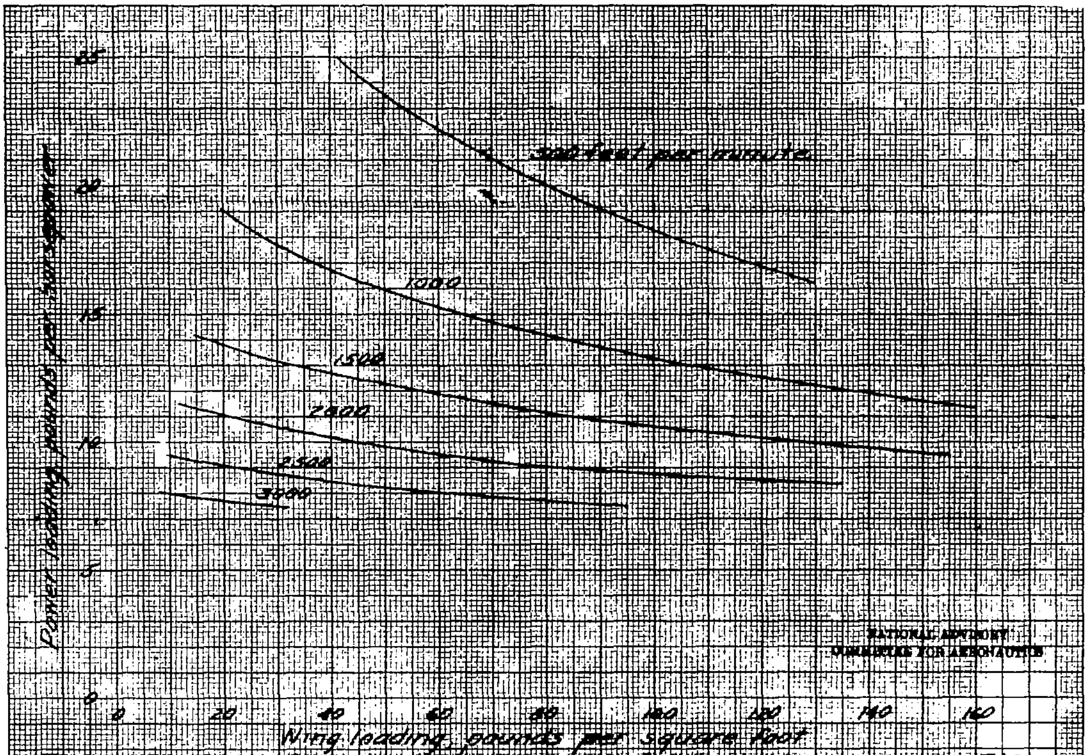

Figure 15 - Rate of climb at sea level with full load. C_{D0} = .0090 + .06 F/S

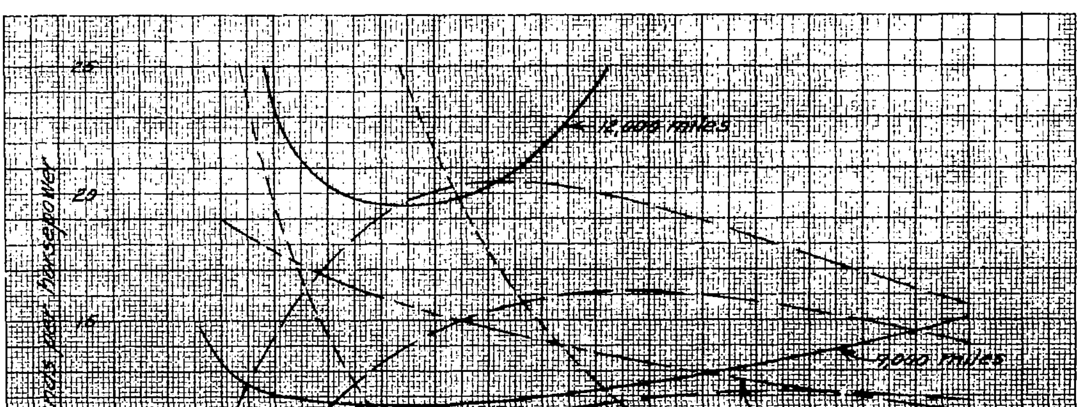

Figure 16 - Performance selection chart C_{D0} = .0090 + .06 F/S

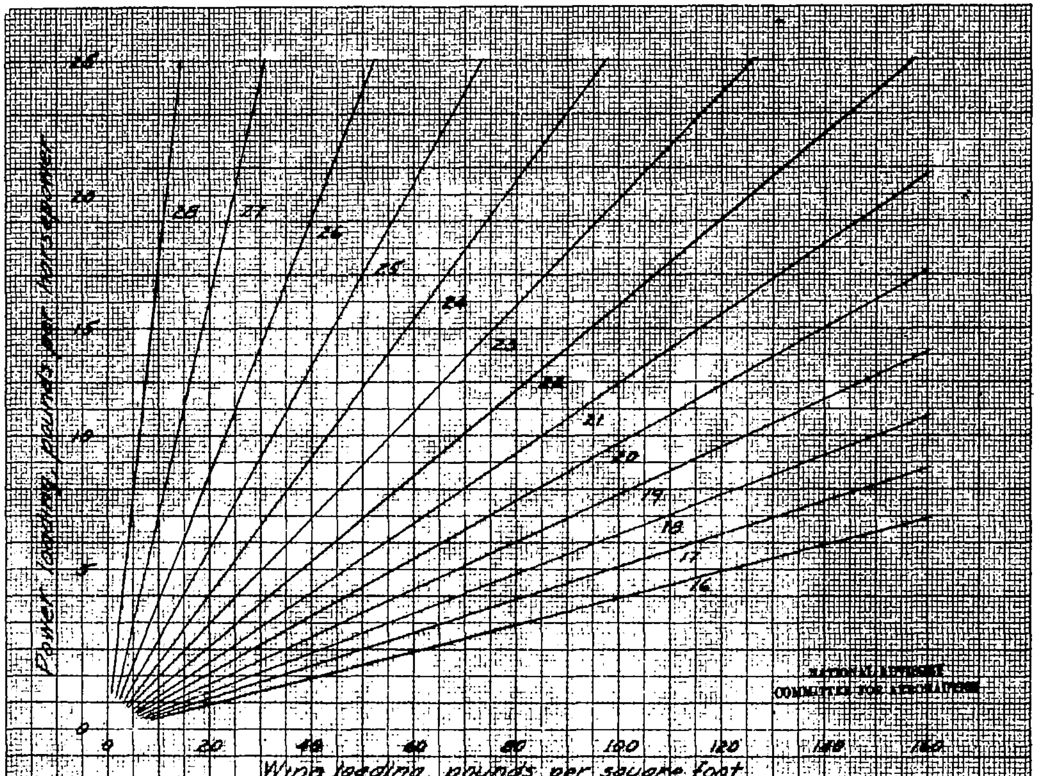

Figure 17.- Maximum lift-to-drag ratio. $C_{D0} = .0090 + 0.06\ F/S$

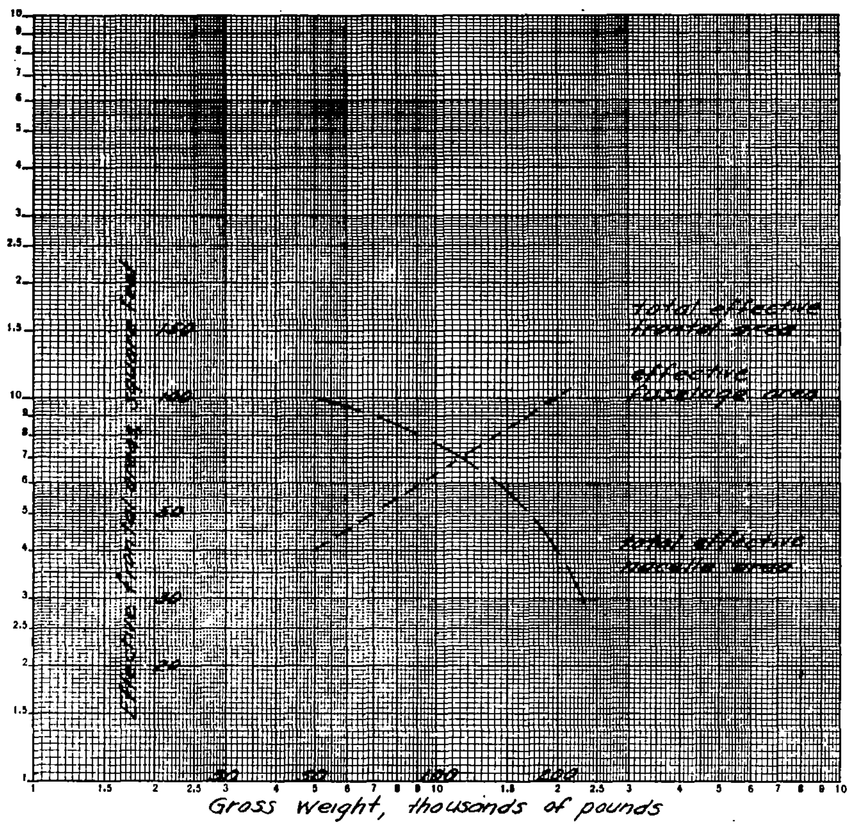

Figure 18 - Variation of effective fuselage and nacelle frontal areas

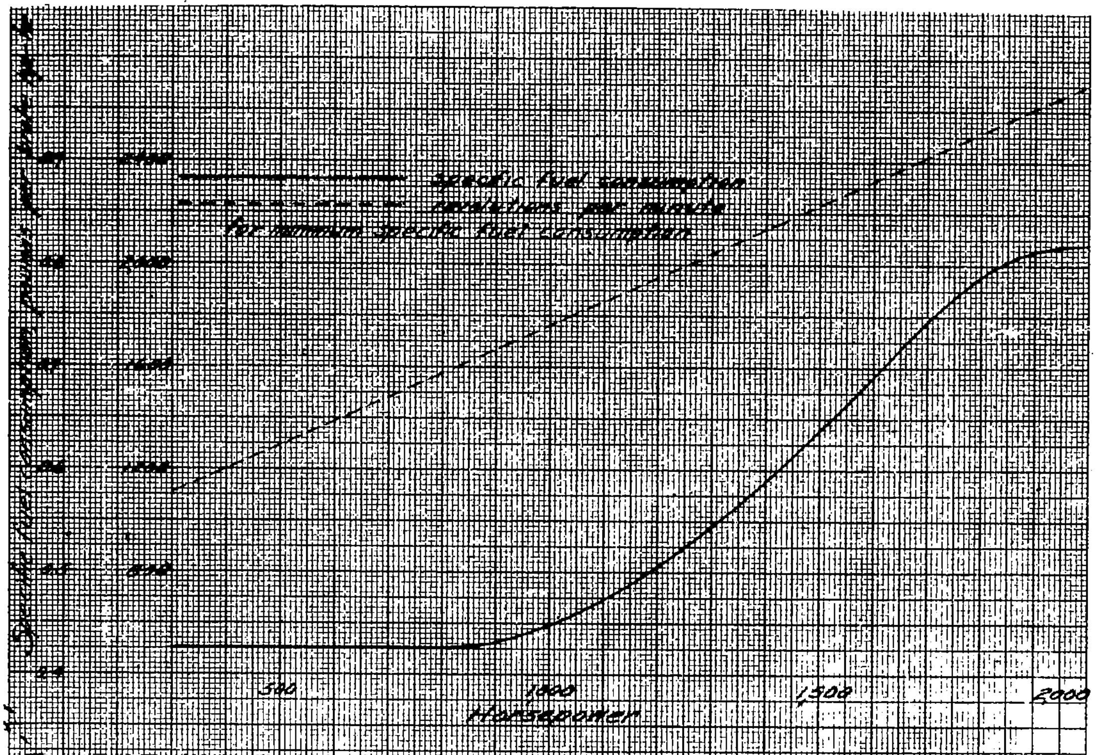

Figure 19 Minimum specific fuel consumption and RPM of 2,000 hp. engine

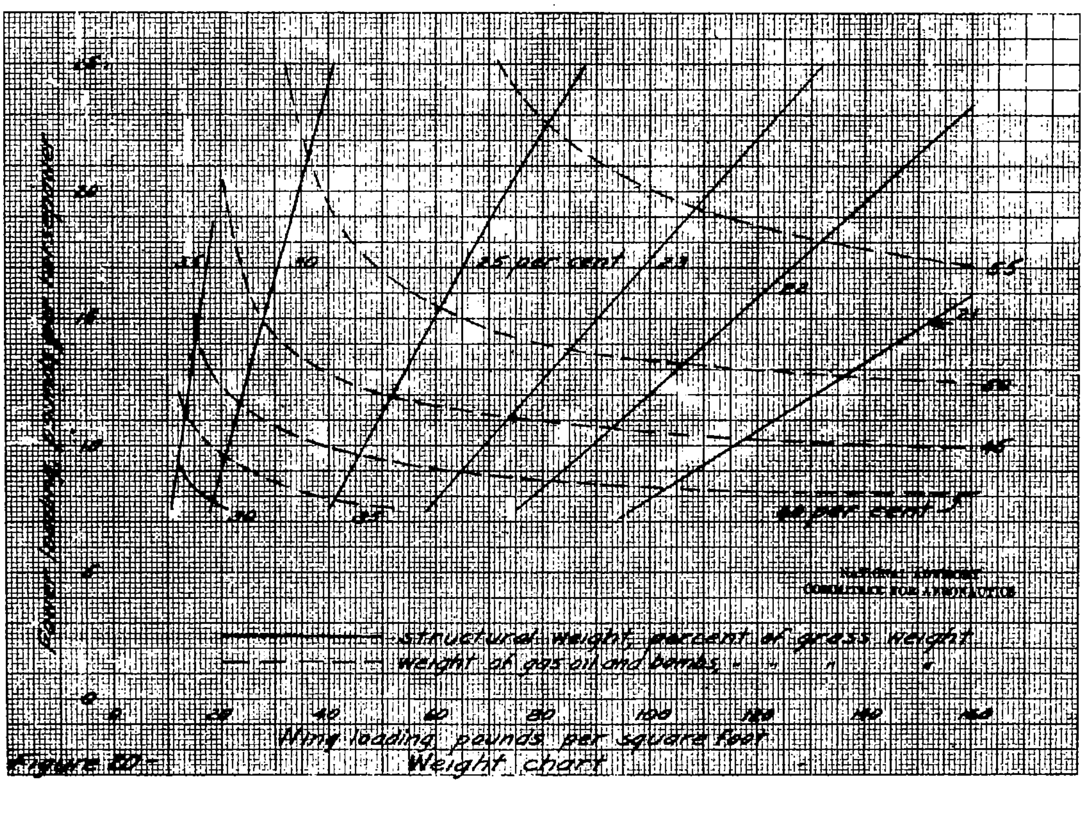

Figure 20 - Weight chart